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THE CONVERSION OF THE ROTOR/WING AIRCRAFT

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SUMMARY

The principal problem associated with the conversion from wing-borne to rotor-borne flight is the possibility of a large attitude disturbance during the first revolution of the rotor. The aircraft disturbance is due to an oscillation of the lift center of pressure in a longitudinal and lateral direction at a frequency that is simply the number of blades multiplied by the rotor rotational speed. The results of a wind-tunnel study indicate that large-amplitude cyclic pitch is one means of eliminating the source of the aircraft disturbance for a three-bladed rotor/wing aircraft. In addition, the selection of four blades on a rotor/wing aircraft may so substantially reduce the disturbing moments that cyclic pitch is not required to eliminate the moments.

The results of an analytical investigation indicate the disturbance may also be reduced in magnitude by a rapid initial rotor acceleration or by the use of an oscillation of the horizontal tail surfaces (elevons) at the frequency of the lift center-of-pressure oscillation. However, simultaneous use of the elevons for longitudinal and lateral control will increase the minimum conversion speeds.

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INTRODUCTION

Recent efforts in the United States on the composite-lift aircraft have been made in an attempt to combine the reduced downwash effects and high hovering efficiency of the helicopter with the high cruise speed efficiency of the conventional fixed-wing aircraft. Most of the composite-lift aircraft are based on independent lift systems, where an attempt can be made to optimize the hovering and cruising lift systems individually.

One system, the rotor/wing, combines the hovering and cruising lift systems into a single lifting surface, a large center hub with three hingeless blades, in an attempt to reduce the weight penalty associated with independent lift systems. For vertical flight this single lifting surface rotates, while for aircraft flight the rotor is stopped with one blade forward in the direction of flight.

The purpose of this paper is to examine the aircraft aerodynamics and dynamics that may be encountered with a rotor/wing aircraft during conversion. It has often been pointed out (for example, see ref. 1), that the rotor/wing composite-lift aircraft will experience unacceptable attitude disturbances during attempts to either start or stop the rotor if all lift is maintained on the rotor. In this paper the results of wind-tunnel and analytical investigations are used to analyze the motions of this composite-lift aircraft during the conversion. The effect of configuration changes, rotor acceleration rate, and control manipulation are evaluated for the critical phase of the conversion.

GENERAL CONSIDERATIONS

The initial technical concern with the feasibility of the rotor/wing aircraft has been directed at the possibility of an attitude disturbance during the starting or stopping of the rotor. Two model tests at NASA Langley have investigated this possibility. The first model test established the magnitude of the disturbing moments and determined that variations in the rotor planform are not significant in reducing the disturbing moments for a three-blade rotor/wing. The second model test established that large-amplitude cyclic blade pitch on a three-blade configuration or the choice of a four-blade configuration can essentially eliminate the disturbing moments.

This paper is organized to: describe the concept, summarize the test procedure and results of the NASA wind-tunnel investigations, present an analysis of the aircraft behavior assuming no attempt at controlling the disturbing moments, and to present several methods of reducing either the conversion attitude disturbance or the disturbing moments.

THE CONCEPT

The rotor/wing configuration under study is shown in figure 1. It combines the hovering and cruising lift systems into a single large hub with three hingeless blades. During vertical or helicopter flight the entire assembly rotates as a hot-cycle rotor, while for airplane flight the rotor is stopped with one blade forward as shown in the figure.

Aircraft Description

The composite-lift configuration shown in figure 1 is similar to a recent entry in the U.S. Army Composite Aircraft Program. The subsequent analysis of the conversion of the aircraft will use the physical parameters of the vehicle proposed to the U.S. Army: a design gross weight of 19,635 lbf (87,336 newtons), a wing loading of 37.3 lbf/ft^2 (1786 newtons/meter^2), and an airplane span of 45 feet (13.7 meters). The rotor diameter is 50 feet (15.24 meters) with a hovering disk loading of 10 lbf/ft^2 (478.8 newtons/meter^2). Additional physical parameters of the rotor/wing aircraft are included in the appendix of this paper.

Control

The rotor/wing aircraft uses conventional helicopter and airplane flight controls. The outer 45 percent of the rotor is a conventional hingeless rotor blade with provisions for both collective and cyclic pitch control. Aircraft pitch, roll, and height control are obtained from these controls. Yaw control is obtained with a small fan inserted into the vertical tail surfaces. The controls are designed to provide satisfactory helicopter flying qualities under MIL-H-8501A (ref. 2).

Control in airplane flight is obtained from a conventional elevon and rudder arrangement. The sizes of the fixed and movable control surfaces were chosen to provide satisfactory airplane flying qualities under MIL-F-8785 (ASG) (ref. 3).

Control in autogyro flight and during the conversion is obtained by a combination of the fixed-wing and rotary-wing controls. The controls are interconnected until the rotor is completely stopped.

Conversion

The method of conversion from helicopter to fixed-wing lift is to accelerate the aircraft in the rotary-wing mode to an airspeed above the fixed-wing stall speed and then decelerate the rotor until it stops with one blade forward. The entire conversion procedure from rotor-borne to wing-borne flight is shown in figure 2. The aircraft takes off, hovers, and flies at speeds up to approximately 100 knots in the helicopter mode. At a speed near the maximum obtainable in the helicopter configuration the rotor/wing converts to autorotational flight and a further increase in speed is obtained from the airplane means of propulsion. At a speed greater than the stall speed of the airplane configuration the rotor is slowed, stopped, positioned, and locked in the airplane mode. At speeds above the conversion speed the aircraft flies as a conventional airplane. The procedure is reversed for the wing-borne to rotor-borne conversion. The conversion plan has been documented in detail in reference 4.

MODEL TESTS

Two wind-tunnel investigations were performed by NASA Langley to document the characteristics of the rotor/wing composite-lift aircraft. The first investigation was performed at the request of the U.S. Navy and with the cooperation of the Hughes Tool Company, the originator of this particular composite-lift aircraft. The first investigation was conducted with a rotating-wing model. The second investigation was initiated by NASA Langley in an attempt to solve the problems defined in the first investigation. The model used in the second investigation, for the sake of simplicity, was a nonrotating model.

Rotating-Wing Model

Rotating-wing model.- The objective of the first investigation was to determine the magnitude of disturbing moments during rotor starts and stops and to determine the effect of variations in the rotor/wing planform on the disturbing moments. The models tested were rigid and were mounted on a relatively soft six-component wind-tunnel balance. Cyclic and collective blade pitch control was provided on the rotors for pitch, roll, and thrust control. For the test results provided in this paper, the models were tested without a horizontal tail.

Test procedure with rotating-wing model.- The composite-lift conversion was investigated using two techniques. For both techniques the tunnel velocity was constant. In the first technique, the operating conditions were defined with the model powered at various fixed rotor speeds from 100 percent to 15 percent of the normal rotor speed. In the second technique, the rotor acceleration (increasing rotor speed from zero to normal helicopter rotor speed) and the rotor deceleration (return to zero rotor speed) were made without power to the model.

The first technique, which may be described as a pseudo-conversion, was used to determine the zero torque condition which represents the boundary between the rotor accelerating conditions and decelerating conditions. The rotor speed was set, the angle of attack of the model was varied, the collective blade pitch was adjusted to maintain the scale lift, and the model power required was noted. The mean rolling moment was trimmed to zero with cyclic blade pitch though no attempt was made to trim the mean pitching moment to zero since it was assumed that the horizontal tail surface would provide the necessary trim control. Based on these data, a schedule of the model angle of attack, collective blade pitch, and cyclic blade pitch as a function of rotor speed was obtained for the conversion.

In the second technique, using the boundary determined from the pseudo-conversion, the conversions were then performed without power to the model. The model angle of attack, collective blade pitch, and cyclic blade pitch were manually adjusted by noting the actual rotor speed and adjusting the controls to match the predetermined schedule. A separate operator was assigned to each control.

The initial or final conditions (i.e., the zero rpm condition) for all conversions were the conditions of zero cyclic and collective blade pitch. The rotor acceleration was initiated by lowering the collective from zero to negative angles to produce an accelerating torque on the rotor. The rotor deceleration was initiated by setting a large positive collective pitch to produce a decelerating torque on the rotor. As the rotor speed approached zero, the collective pitch was reduced to or near zero as the rotor stopped.

Test results with rotating-wing model.- The most significant characteristic observed during the conversion was an oscillation of the lift center of pressure both longitudinally and laterally, at a frequency equal simply to the number of blades times the rotor rotational speed. A typical time history of this oscillation during a rotor start and rotor stop is shown in figure 3. The oscillation is the predominant characteristic found throughout the rotor speed range during the unpowered conversions. A mean longitudinal trim shift equal to one-half of the peak-to-peak amplitude of the longitudinal moment oscillation is also evident. Suprisingly, there was no significant variation in the mean lift at constant angle of attack during the initial rotor start or stop.

The longitudinal and lateral oscillations of the lift center of pressure are due to the rotation of the center of lift in an elliptical path on the rotor hub as illustrated in figure 4. The instantaneous location of the center of lift is plotted on the rotor/wing planform for azimuth positions corresponding to one-third of a revolution of the rotor. The blade to which the azimuth notation refers is shaded. The $\psi = 60^\circ$ azimuth position corresponds to the wing position for airplane flight. The dotted line indicates the path the lift center of pressure follows, while the heavy point indicates the instantaneous location of the center of lift. Because of the symmetry of the model, the center-of-pressure variation repeats itself three times each revolution.

There was no significant difference in the amplitude of the center-of-pressure oscillation due to varying the hub planform. The initial and final amplitudes of the pitching- and rolling-moment oscillations for three rotor/wing planforms are tabulated in figures 5 and 6. The peak-to-peak values of the pitching and rolling moments are presented as a lift center-of-pressure fluctuation in percent of the rotor radius. The data which were obtained from seven rotor-start and rotor-stop cycles indicates that there is no clear advantage between configurations on the basis of the variation of longitudinal or lateral lift center of pressure during the first revolution.

Preliminary calculations, based on the test results obtained with the rotating-wing model, indicated that any significant attitude disturbances from the center-of-pressure oscillation would occur at rotor speeds below about 10 percent of the normal helicopter rotor speed. At such low rotor speeds it was determined that the rotor/wing forced response phase angle was considerably less than 20° of blade azimuth, suggesting that methods of center-of-pressure control could be studied with static models. Based on these conclusions, the second wind-tunnel investigation was initiated.

Nonrotating Model

The object of the second wind-tunnel investigation was to determine what mechanism could be used to control the lift center of pressure during a "rotation" of the rotor wing. The significant results were that cyclic blade pitch on a three-blade configuration will eliminate significant lift center-of-pressure movement, and that there is no significant center-of-pressure motion on a four-blade configuration.

The test technique utilized to investigate the conversion of this composite-lift configuration was to successively rotate and lock the wing in various azimuth positions. At each new blade azimuth position the forces and moments were measured for an angle-of-attack range. With a set of successive rotations of the rotor, the static variation of forces and moments could be obtained corresponding to that experienced by a very slowly rotating rotor.

The models tested had flat-plate three- and four-blade rotor/wing planforms. Both planforms had the same blade-span-to-radius ratio. During a major portion of the tests with the three-blade configuration the blade pitch was scheduled with azimuth position in an attempt to eliminate lift center-of-pressure movement as a function of azimuth position. The blade pitch angle schedules were conventional helicopter cyclic pitch.

Center-of-pressure movement on three-blade rotor.- The effect of large-amplitude cyclic blade pitch on the lift center-of-pressure movement during one-third of a rotor revolution is shown in figure 7. The data presented are for a blade pitch input of $\theta = 0$ and for $\theta = -30 \sin \psi$. The case for zero cyclic blade pitch is seen to produce essentially the same three-per-rotor-revolution oscillation as previously measured during actual rotor starts and stops as shown in figure 3. For the case with which the cyclic pitch was used, however, the longitudinal variation of the center of pressure during the rotor revolution was greatly reduced, and the lateral variation of the center of pressure was partially reversed. The near elimination of the longitudinal center-of-pressure oscillation and the reversal and reduction of the lateral center-of-pressure oscillation suggests that virtually complete elimination of the lift center-of-pressure oscillation could be obtained.

The amplitude of cyclic blade pitch required to reduce the lift center-of-pressure oscillations is reduced at lower aircraft angles of attack. The effect of angle of attack and cyclic blade pitch on the peak-to-peak moments due to the lift center-of-pressure oscillations is presented in figure 8. The cyclic blade pitch appears to be more effective in changing the amplitude of the peak-to-peak rolling moment at low angles of attack. However, 15° of cyclic blade pitch affects the peak-to-peak pitching moments by nearly a constant amount up to approximately 12° angle of attack. The loss of cyclic pitch effectiveness above 12° angle of attack and the flattened peak-to-peak rolling-moment curve above about 10° angle of attack is believed to be due to stall or separation on the flat-plate blades and hub.

An increase in the rotor-wing thickness and blade camber might result in a further increase in the amplitudes of the peak-to-peak moment coefficient beyond the angles of attack for flow separation shown in figure 8; however, the result for the case of zero blade angle would be primarily in the maintenance of a constant amplitude of the lift center-of-pressure oscillation. Considering all effects suggests that the lift center-of-pressure oscillation during conversion may be less of a problem at higher conversion airspeeds, where, because of the lower angles of attack, considerably smaller cyclic pitch inputs may be required to reduce the center-of-pressure oscillation.

Lift penalty with three-blade rotor.- A lift penalty is associated with the requirement for large amplitude cyclic pitch at high rotor/wing angles of attack to reduce the rotor wing center-of-pressure oscillations to an acceptable magnitude. The variation of lift coefficient with angle of attack with and without cyclic blade pitch is presented in figure 9. The solid and broken lines indicate the lift coefficient of the nonrotating wing with and without cyclic blade pitch, respectively. The dotted areas indicate the amplitude of the lift-coefficient oscillations during a rotor revolution for zero cyclic pitch, and the lined area indicates the amplitude of the lift-coefficient oscillations for 30° of cyclic blade pitch. For the range of interest, the mean lift coefficient is reduced approximately one-tenth by the introduction of large-amplitude cyclic pitch to reduce the center-of-pressure oscillation. However, the amplitude of the lift oscillation is reduced. Generally, at constant angle of attack, the lift oscillation would result in approximately a plus-and-minus one-tenth g vertical oscillation of the aircraft at the frequency of three times the rotor speed.

Effect of number of blades.- The selection of four blades on this composite-lift design may completely eliminate any concern for aircraft motions due to the lift center-of-pressure oscillation. The effect of the number of blades on the magnitude of the peak-to-peak moments during a rotor revolution is shown in figure 10. Both of the models had the same diameter, but the effective solidity and wing area of the four-blade rotor/wing is substantially increased for the test model. Examination of the data for the four-blade configuration indicates that the longitudinal center-of-pressure oscillation is about plus-and-minus 2 percent of the rotor radius and that the lateral center-of-pressure oscillation is about plus-and-minus 0.75 percent of the rotor radius at an angle of attack of 12°. Since the four-blade configuration is not an optimum shape, these results indicate that further efforts to define an ideal rotor/wing shape may prove fruitful.

ANALYSIS OF AIRCRAFT BEHAVIOR DURING CONVERSION

The effect of a lift center-of-pressure oscillation during conversion will be shown to produce an initial pitch and roll attitude disturbance during the first or last revolution of the rotor. The amplitude of the disturbance is proportional to the amplitude of the lift center-of-pressure oscillation.

In order to evaluate the significance of the lift center-of-pressure oscillation, a two-degree-of-freedom analysis was made of the motions of a typical three-blade composite-lift aircraft. The physical and aerodynamic characteristics are those of the aircraft design proposed for the U.S. Army Composite Aircraft Program. The analysis assumed that the three-per-revolution lift center-of-pressure oscillation was of constant magnitude and phasing during the rotor start for rotor speeds from zero to 20 percent of the normal rotor speed. The analytical method contained in the appendix can be applied to any number of blades.

Pitch Attitude Disturbance

An initial pitch attitude change of 0.10 radian is shown to occur during the first half-revolution of a rotor start with this composite-lift aircraft. A time history of the calculated longitudinal response of this composite-lift aircraft is presented in figure 11 for a rotor start at an airspeed of 150 knots. For the case presented, it was assumed that the longitudinal three-per-revolution center-of-pressure oscillation had a peak-to-peak amplitude of 25 percent of the rotor radius (CP_Y = 12.5) and the lateral oscillation had a peak-to-peak amplitude of 12.5 percent of the rotor radius (CP_X = 6.25). The airplane damping in pitch and roll was augmented by additional rotor damping and gyroscopic terms proportional to rotor speed. A longitudinal trim change equal to one-half of the peak-to-peak pitching moment was considered to have been applied by the elevator control in order to retrim the mean pitching moments on the aircraft to zero during the first half-revolution. Based on a calculation of the angular acceleration of the rotor during an autorotation start, the rotor angular acceleration was assumed to be constant at 1.6 radians/second^2.

The peak pitch attitude disturbance of 0.10 radian occurs simultaneously with completion of the first half-revolution of the rotor. At the assumed rotor acceleration, the first half-revolution of the rotor is obtained in 1.6 seconds. After the initial spike, the amplitude of the pitching rate oscillations reduces with increasing rotor speed resulting in little further attitude change. The pitch attitude, after the initial disturbance, returns toward the initial trim point under the influence of the aircraft static stability.

Roll Attitude Disturbance

An initial roll attitude change of 0.20 radian will occur during the first one-third of a revolution of a rotor start with this composite-lift aircraft. The time history of the calculated lateral response during a rotor start is presented in figure 12. The same conditions and assumptions apply to the lateral response as were applied to the longitudinal response. After the initial roll rate spike, the amplitude of the roll rate oscillations reduces with increasing rotor speed and affects the roll attitude very little. The subsequent roll attitude drift is due to the increasing magnitude of the assumed gyroscopic moment coupled to the mean nose-down pitching velocity.

Significance of Results of Analysis

The only significant attitude disturbance, due to the lift center-of of-pressure oscillation, occurs during the first rotor revolution of the accelerating conversion. A similar effect can be anticipated during the last rotor revolution of a decelerating conversion. The rotor speed at the time of the attitude disturbance is below 10 percent of the normal rotor speed, where the effects of rotor rotation on aerodynamics and structural dynamics are small.

CONTROL OF THE ATTITUDE DISTURBANCE

Attitude disturbances can be reduced in magnitude or possibly eliminated by the use of cyclic blade pitch, use of a four-blade instead of a three-blade configuration, the use of oscillating elevon deflection, or by the use of high rotor angular acceleration and deceleration. Simply accepting an attitude disturbance during conversion is not believed to be practical. From an operational viewpoint, the aircraft should have reasonable maneuver freedom at any point in the conversion. The use of methods to control the lift center-of-pressure position or to counter the resulting pitch and roll moments will prevent a rigid conversion technique from being required. Each of the following possible approaches to prevent a large attitude disturbance must be evaluated by the designer.

Cyclic Blade Pitch

The use of cyclic blade pitch was shown in figure 8 to provide a positive control of the amplitude of both the longitudinal and lateral motions of the lift center of pressure. Some automatic device, operating as a function of rotor speed and angle of attack, might be developed to control the amplitude and phasing of the cyclic blade angle and to return the cyclic pitch control to the pilot for helicopter flight.

Four-Blade Rotor/Wing

The use of a four-blade composite-lift configuration was shown in figure 10 to provide a substantial reduction in the lift center-of-pressure oscillations without the use of cyclic blade pitch. The results with the four-blade configuration examined, while not an optimum shape, do indicate that further efforts to define an ideal composite-lift plan-form may prove fruitful.

Rotor Angular Acceleration

It is shown in the appendix that increasing the rate of acceleration of the rotor is a powerful tool in reducing the amplitude of the attitude upset when no attempt is made to control the center-of-pressure oscillations. This result, and the requirement for indexing the blades at a given position, suggests that a precise and powerful rotor speed control mechanism is required during conversion.

Elevon Control

The elevons may be used to counter the pitch and roll moments produced by the lift center-of-pressure oscillation. The longitudinal and lateral control capability of the aircraft assumed for analysis is adequate for trim above 143 knots.

Longitudinal control.- The longitudinal moment available from the horizontal tail as a function of the airspeed is presented in figure 13. Included is the required moment to balance the longitudinal trim shift and the most nose-up moment during the rotor start. The required moments are based on the assumption that the amplitude of the longitudinal center-of-pressure oscillation was constant throughout the speed range. Since the available longitudinal control is capable of trimming the moments due to the longitudinal center-of-pressure oscillation, there is some control margin available for maneuver requirements. That is, it is possible to eliminate the oscillatory longitudinal response during the rotor start by using a pitch oscillation of the horizontal tail and still retain some maneuver margin.

The initial pitch attitude disturbance might be controlled by a single pitch motion of the elevons; however, vibrational inputs subsequent to the first half-revolution of the rotor may be too extreme for passenger comfort. Therefore, it appears reasonable to require a pitch oscillation of the elevons during the rotor start for rotor speeds from 0 to 10 percent of the normal helicopter rotor speed.

The horizontal tail deflections required to completely trim the rotor/wing pitch moments during a rotor start up to 10 percent of normal rotor speed are presented in figure 14, assuming no time lag between the control input and the control moment. If the horizontal tail oscillation is stopped at the end of the second rotor revolution, as shown, the longitudinal center-of-pressure oscillation would then result in about a plus-and-minus 0.025 radian/second pitching velocity oscillation of the aircraft. (See fig. 11.) The aircraft pitch attitude changes associated with these pitching velocities would then be negligible.

Control of the horizontal tail oscillations required to counter the longitudinal center-of-pressure oscillation may be an extremely taxing design problem. An automatic device would appear to be necessary in that, if attempted manually, the pilot could get out of phase and cause an aircraft upset.

Lateral control.- The lateral control moment available is adequate to trim out the lateral center-of-pressure oscillation above an airspeed of 143 knots. The lateral control moment available and that required to trim out the lateral center-of-pressure oscillation is presented in figure 15. The available control is based on the assumption that the elevons are differentially deflected to 30° with no requirement for roll control due to yaw or for loss of roll effectiveness due to elevator deflection. There is no margin for maneuvering at this speed. Simultaneous use of the elevons for longitudinal and lateral trim of the center-of-pressure oscillation and for maneuvering will increase the minimum conversion speeds above 143 knots.

Aeroelastic Effects

The analysis of aircraft motions due to the lift center-of-pressure motion, and the choice of methods of control of the attitude disturbance is based on data from rigid models. It appears that aeroelastic effects would increase the amplitude of the center-of-pressure motion and further aggravate the attitude disturbance problem during conversion. The possibility of larger amplitude center-of-pressure motion, when aeroelastic effects are considered, suggests that solutions which attempt to control the center-of-pressure motion should be favored.

CONCLUDING REMARKS

The data, analysis, and discussion contained in this paper will provide a basis for determining the direction in which future research on the rotor/wing aircraft may proceed. The material presented is not intended to provide a final solution to the problems encountered in conversion with the rotor/wing aircraft. The aeroelastic problems associated with the conversion must also be considered. Although this paper has not attempted such a treatment, examination of solutions to the attitude disturbance problems should not be considered complete without considering the effect of the solutions on the aeroelastic problems.

On the basis of the data and analysis included in this paper, the following tentative conclusions may be drawn.

1. The principal problem associated with the conversion from wing-borne to rotor-borne flight of a three-blade rotor/wing is the possibility of a large attitude disturbance during the first revolution of the rotor.

2. The aircraft disturbance is due to a rotation of the lift center of pressure in an elliptical path at a frequency that is simply the number of blades times the rotational speed.

3. The moments causing the longitudinal and lateral disturbance of a three-blade rotor/wing can be trimmed by an oscillation of the horizontal tail surface (elevons); however, simultaneous use of the elevons for trim and maneuvers will increase the minimum conversion speeds.

4. Large-amplitude cyclic pitch is one means of eliminating the source of the aircraft disturbance on a three-blade configuration.

5. The adoption of a four-blade configuration on a rotor/wing aircraft may result in sufficiently small disturbing moments during the start or stop that concern for an attitude disturbance will be eliminated.

APPENDIX

Symbols

b	wing span, ft (m)
c	wing chord, ft (m)
C_L	lift coefficient, $\frac{L}{qS}$
C_{L_α}	rate of change of lift coefficient with angle of attack, 1/rad
C_l	rolling-moment coefficient, $\frac{M_X}{qSb}$
C_{l_p}	damping coefficient due to rolling velocity, $\frac{\partial C_l}{\partial \frac{pb}{2V}}$
C_m	pitching-moment coefficient, $\frac{M_Y}{qSc}$
C_{m_q}	damping coefficient due to pitching velocity, $\frac{\partial C_m}{\partial \frac{qc}{2V}}$
C_{m_α}	rate of change of pitching-moment coefficient with angle of attack, 1/radian
CP_X	coefficient of $\sin 3\psi$ in expression for lateral center-of-pressure oscillation
CP_Y	coefficient of $\cos 3\psi$ in expression for longitudinal center-of-pressure oscillation
D_q	damping moment proportional to and opposing aircraft pitching velocity, lbf-ft/radian/sec (N-m/radian/sec)
D_p	damping moment proportional to and opposing aircraft rolling velocity, lbf-ft/radian/sec (N-m/radian/sec)
E	multiplying coefficient of rotor contribution to aircraft angular velocity damping and gyroscopic cross coupling (set equal to zero to eliminate their contribution to and to decouple longitudinal and lateral degrees of freedom)
H_q	gyroscopic moment proportional to aircraft rolling velocity, lbf-ft/radian/sec (N-m/radian/sec)
H_p	gyroscopic moment proportional to aircraft pitching velocity, lbf-ft/radian/sec (N-m/radians/sec)

I_V blade mass moment of inertia about virtual hinge, slug-ft^2 (kg-m^2)

K_α spring moment proportional to and opposing a change in the aircraft pitch attitude, lbf-ft/radian (N-m/radian)

k_Y mass radius of gyration of aircraft about X axis, ft (m)

k_X mass radius of gyration of aircraft about Y axis, ft (m)

L lift

M mass of aircraft, slugs (kg)

M_A longitudinal aircraft moment, lbf-ft (N-m)

M_B lateral aircraft moment, lbf-ft (N-m)

$M_X(t)$ time variant rolling moment, lbf-ft (N-m)

$M_Y(t)$ time variant pitching moment, lbf-ft (N-m)

N_o stick-fixed neutral point, percentage of wing chord

p aircraft rolling velocity, radians/sec

$\dot{p}$ aircraft rolling acceleration, radians/sec/sec

q aircraft pitching velocity, radians/sec; or dynamic pressure, lbf/ft^2 (newtons/meter^2)

$\dot{q}$ aircraft pitching acceleration, radians/sec/sec

R rotor radius, ft (m)

S wing area, ft^2 (m^2)

V velocity, knots

x_{cg} longitudinal location of aircraft center of gravity, percentage of wing chord

α aircraft angle of attack, deg

Δ increment due to subscript notation

γ_V blade mass factor (Lock number) based on virtual hinge offset

θ blade pitch angle at any azimuth position ψ, deg

ψ	blade azimuth angle measured from downwind position in direction of rotation, deg or radian
Ω	rotor angular velocity, radian/sec
$\dot{\Omega}$	rotor angular acceleration, radians/sec/sec
ω_{1NR}	cantilever first-flapwise-mode natural frequency of nonrotating blade, radian/sec
ω_{1R}	cantilever first-flapwise-mode natural frequency of rotating blade, radians/sec

Subscripts:

$\psi = 60^{\circ}$	condition with blade at $\psi = 60^{\circ}$ azimuth position (or other indicated azimuth position)
q	due to pitching velocity
p	due to rolling velocity

Analysis

An analysis of the significance of the oscillatory moments was made by considering the two-degree-of-freedom response of a typical three-blade rotor/wing aircraft to a rotation of the lift center of pressure at three times the rotor speed. Initially assuming no elevon or other control motions to counter the oscillations, the pitching- and rolling-moment forcing functions were written as:

$$M_Y(t) = LR\left(CP_Y\right)(1 + \cos 3\psi) \tag{1}$$

$$M_X(t) = LR\left(CP_X\right)(+\sin 3\psi) \tag{2}$$

The azimuth angle may be represented as the initial blade position in airplane flight (60°) plus the rotational speed times time, i.e.,

$$\psi = \frac{\pi}{3} + \Omega t \tag{3}$$

Therefore, equations (1) and (2) can be written as:

$$M_Y(t) = LR\left(CP_Y\right)\left[1 - \cos(3\Omega t)\right] \tag{4}$$

$$M_X(t) = LR\left(CP_X\right)\left[-\sin(3\Omega t)\right] \tag{5}$$

Constant angular acceleration of the rotor (about the shaft axis) was assumed, therefore

$$\Omega = \dot{\Omega} t \tag{6}$$

Substituting equation (6) into equations (4) and (5):

$$M_Y(t) = LR\left(CP_Y\right)\left[1 - \cos(3\dot{\Omega} t^2)\right] \tag{7}$$

$$M_X(t) = -LR\left(CP_X\right)\left(\sin 3\dot{\Omega} t^2\right) \tag{8}$$

Because there is a mean longitudinal trim shift when the rotor is started, a term to provide a counter trim moment can be added to the bracketed trim of equation (7). It was assumed the counter trim moment would be introduced during the first half-revolution of the rotor. The equation for the counter trim moment to be added to equation (7) is:

$$-LR\left(CP_Y\right)\left(1 - e^{-\Omega t}\right) \tag{9}$$

Substituting equation (6) into equation (9) and adding the resultant expression to equation (7) gives:

$$M_Y(t) = LR\left(CP_Y\right)\left[e^{-\dot{\Omega} t^2} - \cos(3\dot{\Omega} t^2)\right] \tag{10}$$

The equations of motion used for the analysis were

$$\dot{q} + \frac{D_q}{I_Y} q + \frac{K_\alpha}{I_Y} \int_0^t q \, dt + \frac{H_q}{I_Y} p = M_Y(t) \tag{11}$$

$$\dot{p} + \frac{D_p}{I_X} p - \frac{H_p}{I_X} q = M_X(t) \tag{12}$$

The damping, gyroscopic cross coupling, and static stability margin were expressed in terms of the airspeed and the rotor speed. The values of these stability derivatives and the dimensional quantities of the aircraft under analysis are tabulated below

(1) Damping in pitch:

(a) Contribution due to tail: $\left[C_{m_q}\right]_{Tail} = -9.44$

(b) Contribution due to rotation of rotor:* $\left[\Delta\left(\frac{M_A}{I_V\Omega}\right)_q\right]_{Rotor} = -0.48$

(c) Total damping in pitch: $\frac{D_q}{I_Y} = \frac{6000\dot{\Omega}Et + 740V}{Mk_Y^2}$ 1/sec

(2) Damping in roll:

(a) Contribution due to rotor/wing-tail combination:**
$\left[C_{l_p}\right]_{Rotor/wing\text{-}tail} = -0.25$

(b) Contribution due to the rotating rotor:* $\left[\Delta\left(\frac{M_B}{I_V\Omega}\right)_p\right]_{Rotor} = -0.48$

(c) Total damping in roll: $\frac{D_p}{I_X} = \frac{6000\dot{\Omega}Et + 160V}{Mk_X^2}$ 1/sec

(3) Rotor gyroscopic moment:*

(a) $\left[\Delta\left(\frac{M_B}{I_V\Omega}\right)_q\right]_{Rotor} = -\left[\Delta\left(\frac{M_A}{I_V\Omega}\right)_p\right]_{Rotor} = 1.44$

(b) Gyroscopic moment: $\frac{H_q}{I_Y} = \frac{18{,}000\dot{\Omega}Et}{Mk_Y^2}$; $\frac{H_p}{I_X} = \frac{18{,}000\dot{\Omega}Et}{Mk_X^2}$

(4) Longitudinal static stability: $C_{M_\alpha} = C_{L_\alpha}\left(x_{cg} - N_o\right) = 2.86\left(x_{cg} - N_o\right)$

therefore: $\frac{K_\alpha}{I_Y} = \frac{26V^2\left(x_{cg} - N_o\right)}{Mk_Y^2}$

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*Rotor stability derivatives are based on the analysis of reference 5 for a $\omega_{1NR}/\Omega = 1$, $\omega_{1R}/\Omega = 1.5$, and a $\gamma_V = 3$.

**The roll damping is based on stability derivatives obtained with the nonrotating model described in this paper with the blades in the $\psi = 60^{\circ}$ orientation.

Rotor/Wing Dimensional Quantities Assumed for Analysis

Rotor radius	25 ft (7.62 meters)
Wing area	$0.268\pi R^2$
Wing mean aerodynamic chord	0.63R
Wing span	1.8R
Roll radius of gyration, k_X	0.203R
Pitch radius of gyration, k_Y	0.63R
Rotor polar radius of gyration	0.48R
Rotor weight	0.14 design gross weight
Design disk loading	10 lbf/ft^2 (478.8 newtons/meter^2)
Horizontal tail length	1.2R
Horizontal tail span	1.12R
Horizontal tail area	$0.094\pi R^2$
Design rotor speed	28.8 radians/second

Effect of Rotor Angular Acceleration

The effect of rotor angular acceleration on the disturbance amplitude is presented in figure 16. The attitude peak is presented as the ratio of the peak to the corresponding center-of-pressure coefficient. This presentation is possible because the assumptions of the analysis yield a linear set of differential equations. For the conditions assumed for figure 16, the rotor angular acceleration is a powerful tool in reducing the amplitude of the aircraft disturbance. It would appear, based on the calculated time histories of figures 11 and 12, that a high acceleration rate would only be necessary during the first 2 seconds of the rotor start. A similar conclusion can be inferred for the case of a rotor stop. It would appear that a high deceleration (during the last revolution) would do the most to minimize the attitude disturbance problem.

Effect of Conversion Velocity

The effect of the aircraft velocity on the disturbance amplitudes is presented in figure 17. For a direct comparison, the scales of figures 16 and 17 are the same. It is seen that a substantial increase in the airspeed is necessary to reduce the amplitude of the attitude disturbance. Also, the increased dynamic pressure would cause even the reduced attitudes at higher conversion speeds to produce higher vertical accelerations of the aircraft.

Effect of Static Margin and Rotor Damping Moments

The effect of the aircraft static margin and the magnitude of the assumed rotor angular velocity damping moments is presented in figure 18. Increasing the static margin is seen to be a relatively ineffective method of reducing the amplitude of the attitude disturbance.

The assumed angular velocity damping and gyroscopic moments very slightly increase the amplitude of the pitch attitude disturbance. It appears that the gyroscopic coupling is producing the predominant effect. However, because of the very slight effect and the uncertainty in the magnitude of the damping values, it appears reasonable to consider the attitude disturbance problem as uncoupled. That is, in future analysis of the conversion, the lateral-directional response can be considered separately from the complete longitudinal response.

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2. Anon.: Helicopter Flying and Ground Handling Qualities; General Requirements for Mil. Specification MIL-H-8501A, Sept. 7, 1961.

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4. Harned, M. S., and Head, R. E.: Hot Cycle Rotor/Wing High Speed VTOL Aircraft. Proceedings of 1st National V/STOL Aircraft Symposium, American Helicopter Society, November 3, 4, 1965.

5. Ward, John F.: Exploratory Flight Investigation and Analysis of Structural Loads Encountered by a Helicopter Hingeless Rotor System. NASA TN D-3676, November 1966.

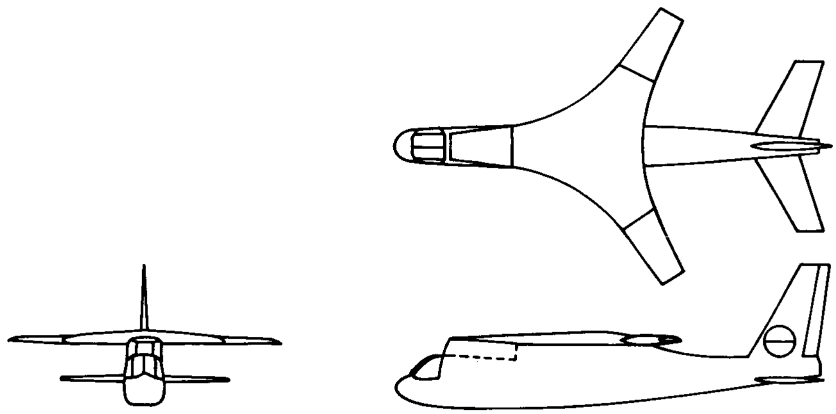


Figure 1.- Typical rotor/wing design.

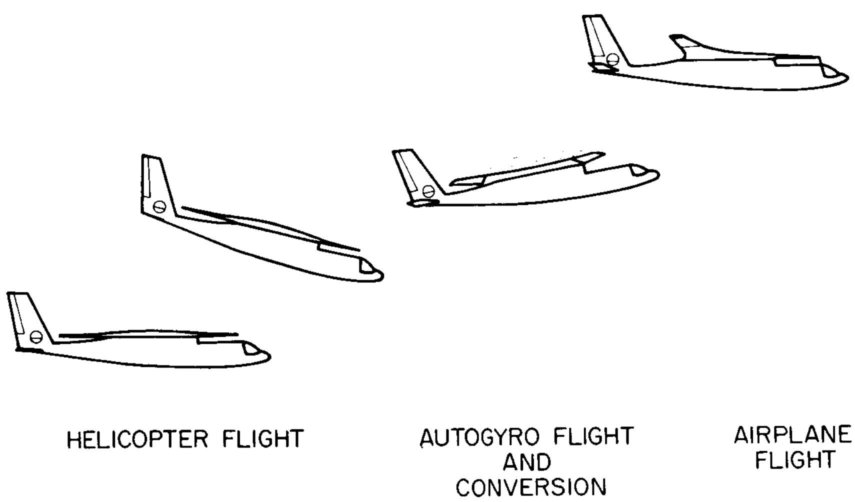


Figure 2.- Rotor/wing flight modes.

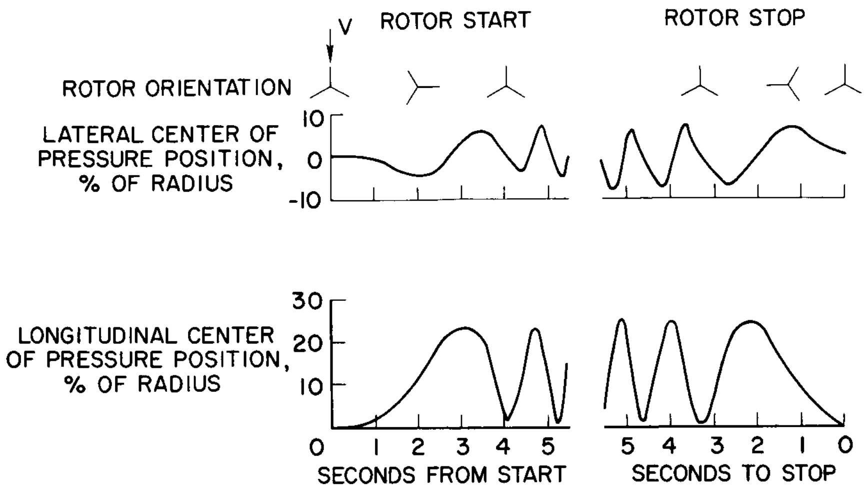


Figure 3.- Typical center-of-pressure movement from airplane condition during rotor starting and stopping.

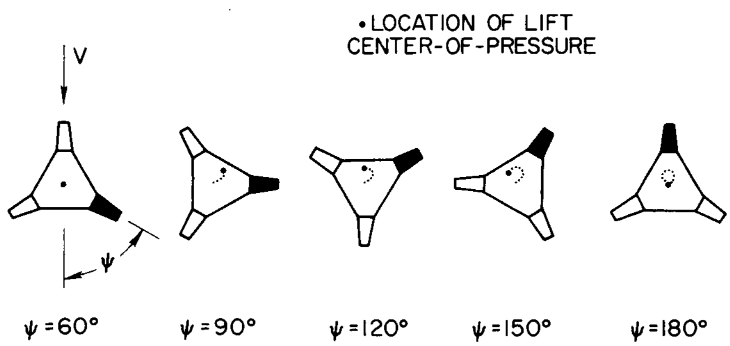


Figure 4.- Lift center-of-pressure motion during rotor revolution.

ROTOR CONFIGURATION	2 CP_Y, % OF R	2 CP_X, % OF R	$\frac{\text{MEAN LIFT}}{q \pi R^2}$	α, deg
1. 	23	12	.22	14.4
2. 	25	15	.22	14.9
	22	13	.21	14.9
	22	15	.23	15.8
	20	13	.22	15.9
3. 	26	13	.18	15.5
	30	12	.18	15.7

Figure 5.- Center-of-pressure oscillation during first revolution of rotor start.

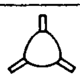
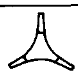
ROTOR CONFIGURATION	2 CP_Y, % OF R	2 CP_X, % OF R	$\frac{\text{MEAN LIFT}}{q \pi R^2}$	α, deg
1. 	25	15	.18	11.4
2. 	27	14	.19	12.7
	24	14	.22	15.9
	31	13	.23	15.8
3. 	27	10	.16	14.3
	33	15	.17	14.3

Figure 6.- Center-of-pressure oscillation during last revolution of rotor stop.

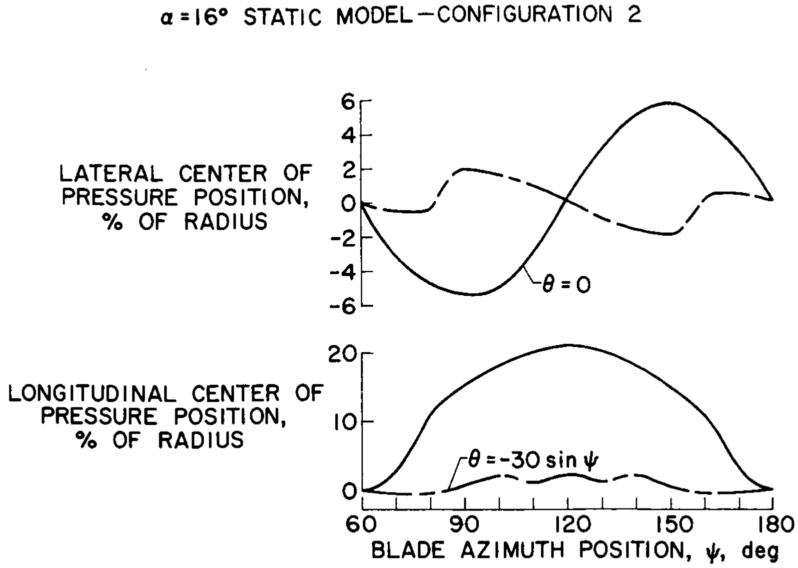


Figure 7.- Center-of-pressure movement from airplane condition during rotor revolution.

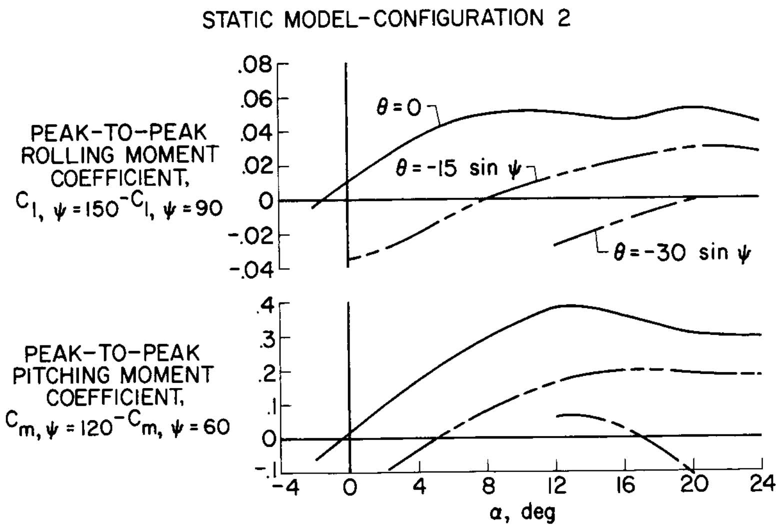


Figure 8.- Effect of angle of attack on moments during rotor revolution.

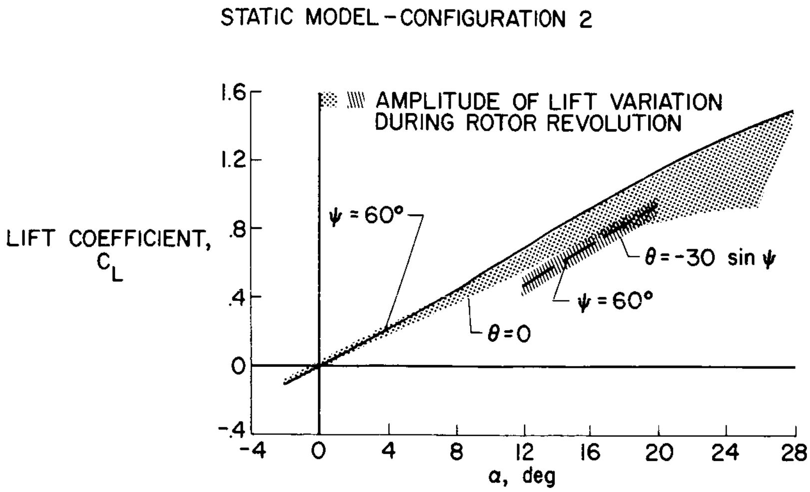


Figure 9.- Effect of large amplitude cyclic pitch on lift.

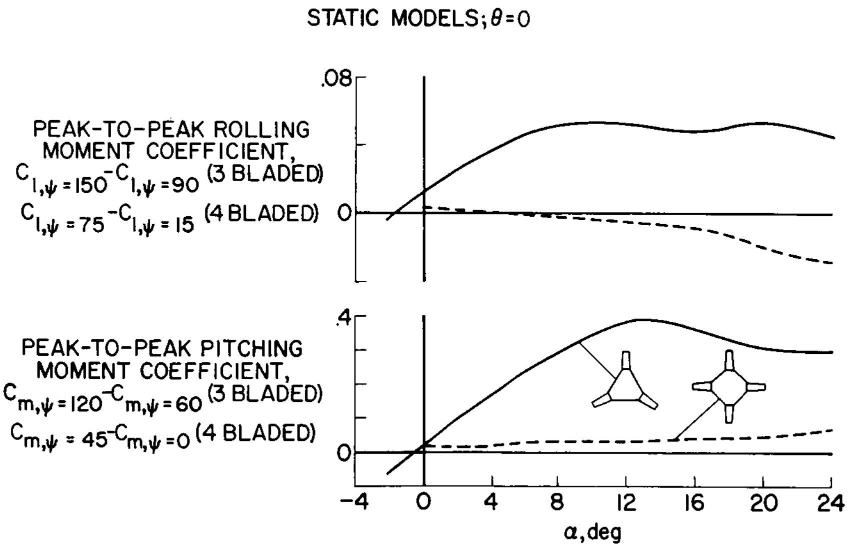


Figure 10.- Effect of number of blades on peak-to-peak moments during rotor revolution.

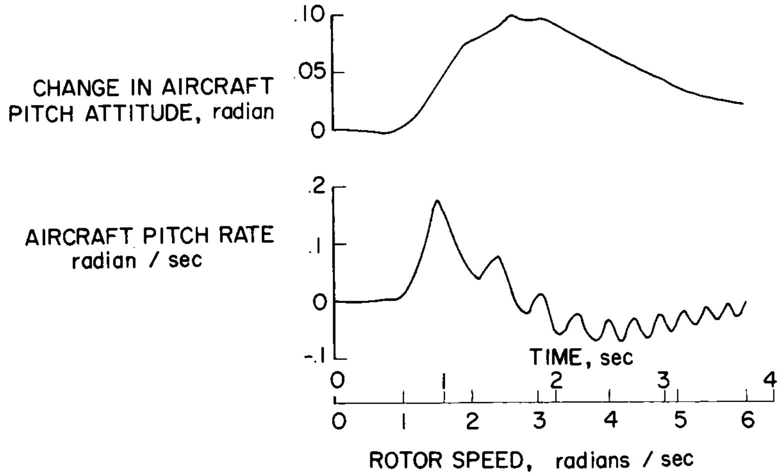


Figure 11.- Longitudinal response of rotor/wing with untrimmed center-of-pressure movement during rotor start.

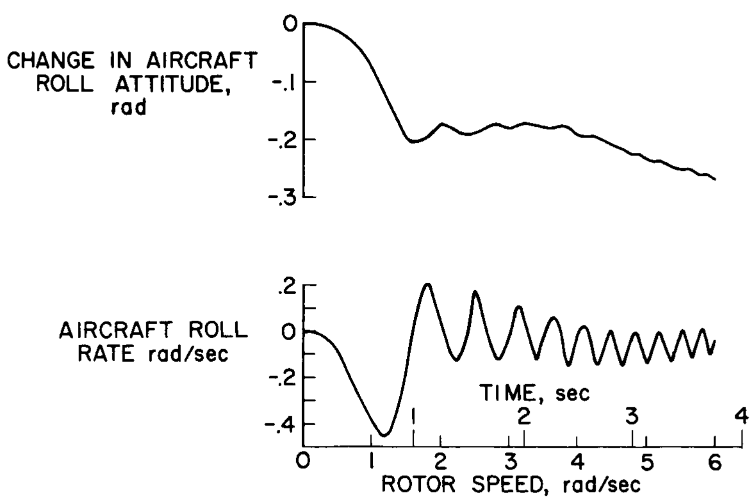


Figure 12.- Lateral response of rotor/wing with untrimmed center-of-pressure movement during rotor start.

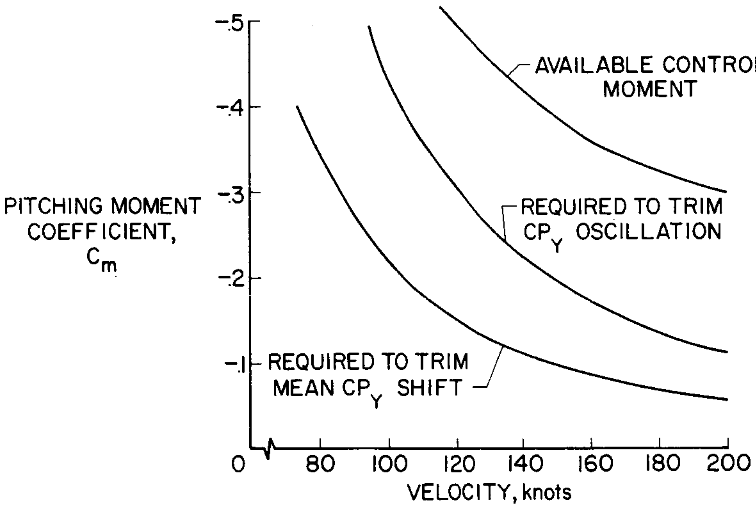


Figure 13.- Rotor/wing longitudinal control.

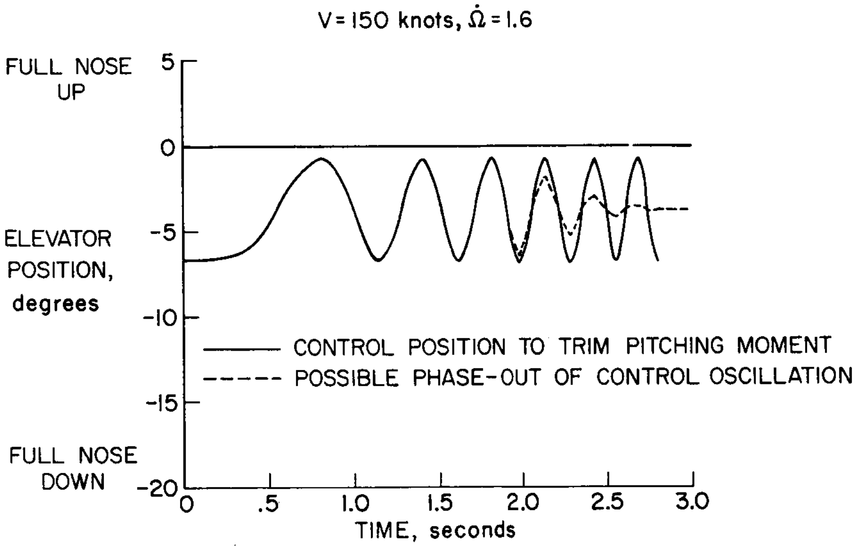


Figure 14.- Horizontal tail deflections to trim longitudinal center-of-pressure oscillations.

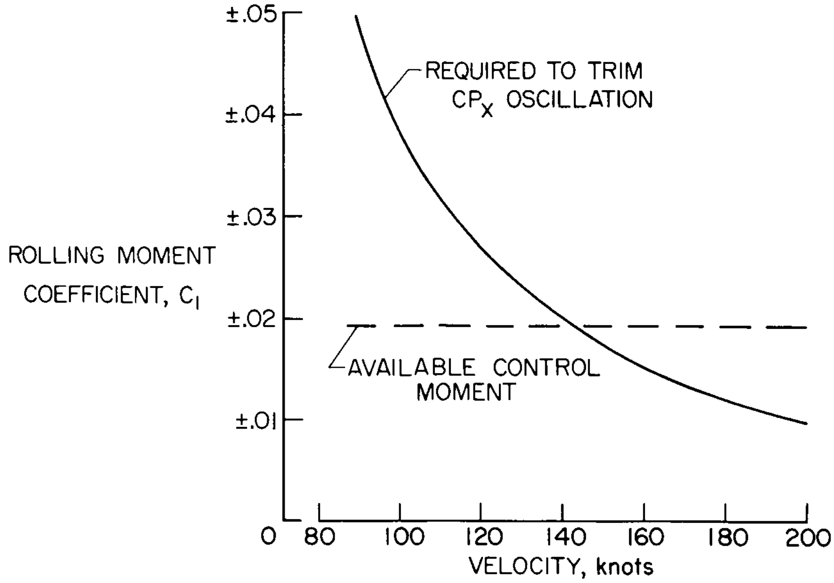


Figure 15.- Rotor/wing lateral control.

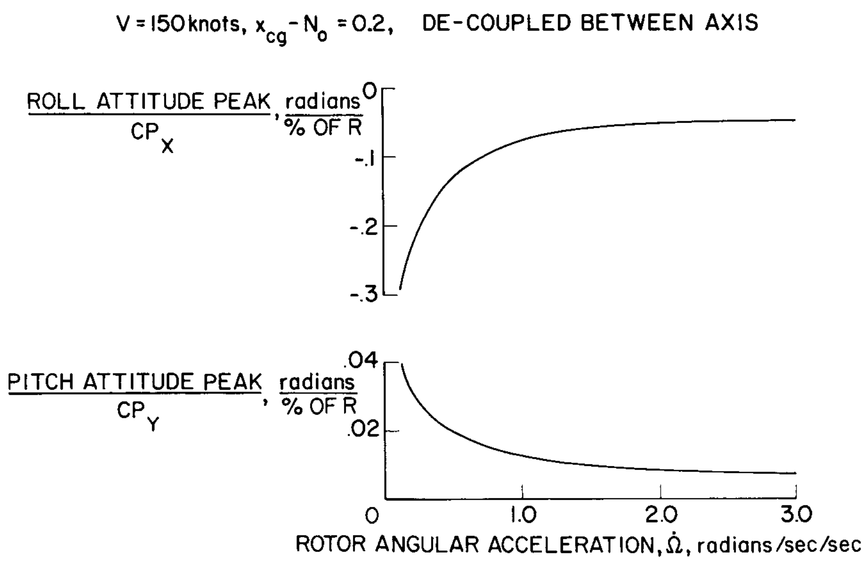


Figure 16.- Effect of rotor acceleration rate on disturbance amplitude during conversion.

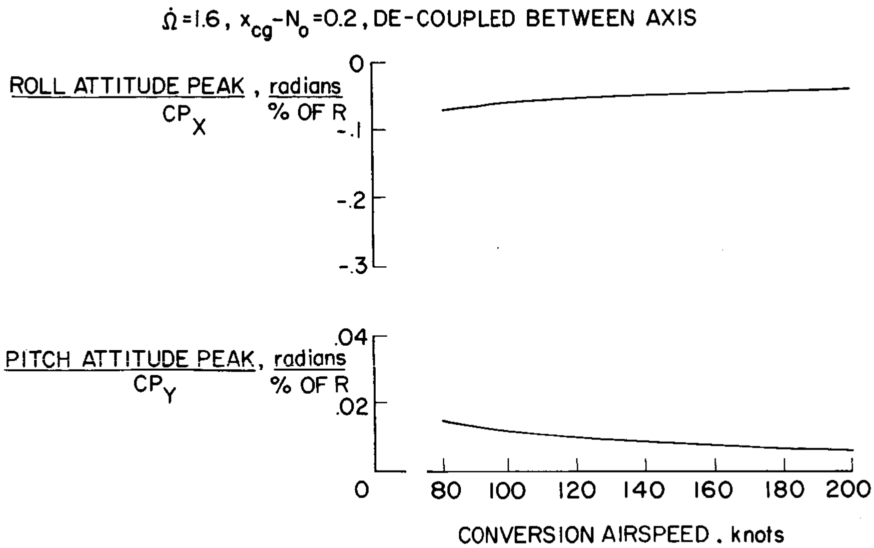


Figure 17.- Effect of airspeed on disturbance amplitude during conversion.

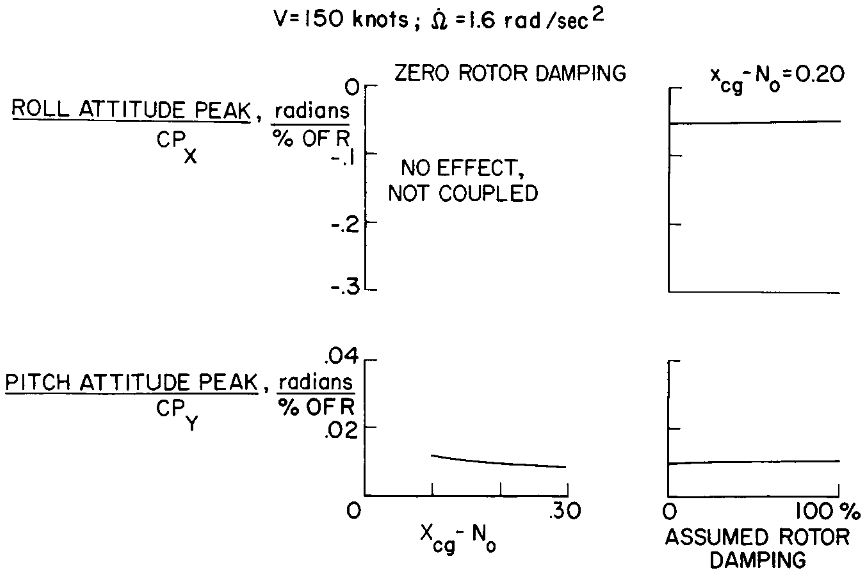


Figure 18.- Effect of static margin and additional rotor damping on disturbance amplitude during conversion.

NASA-Langley, 1967